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DESIGN OF THE THERMAL CONTROL SYSTEM FOR THE SPACE TECHNOLOGY 5 MICROSATELLITE

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ABSTRACT

The New Millennium Program's (NMP) Space Technology 5 (ST-5) Project, currently in Phase B of the design process, is slated to launch three 20-kg class spin stabilized microsatellites in late 2003. The proposed orbit is highly elliptical and could result in an earth shadow eclipse of almost 2 hours. Although ST-5's maximum eclipse is only 2 hours, future missions could involve eclipses as long as 8 hours. As spacecraft size, mass, and available resources decrease and eclipse duration increases, thermal engineers will be challenged to design simple but robust thermal control systems that meet temperature requirements for all phases of the mission. In addition, future similar missions may involve large "fleets" of such small spacecraft, which, for cost and I&T reasons, must be almost identical in design. Such spacecraft will require a generic but robust thermal control design that is suitable for a wide variety of thermal environments. This paper presents the results of a study of three design concepts and preliminary analysis of the design selected for ST-5.

INTRODUCTION

The New Millennium Program (NMP) Space Technology 5 (ST-5) Project will serve as a technology demonstration mission for future NMP and Sun Earth Connection (SEC) Missions. ST-5 endeavors to take advantage of technological achievements to enable the creation of spacecraft constellations that can be launched and managed at an affordable price with acceptable risk. More specifically the goals of the ST-5 Project are to:

1. Design, develop, integrate, and operate a full-service 20-kg class spacecraft through the use of multiple new technologies (Table 1 NMP Technologies).
2. Achieve accurate research quality scientific measurements using this class of spacecraft[1].
3. Design, develop, and operate multiple spacecraft to act as a single constellation rather than as individual elements.

Table 1 NMP Technologies

	Technology	Provider
Spacecraft Miniaturization Technologies	Li-Ion Battery	AEA Technology
	Multifunctional Structures	Lockheed-Martin
	Variable Emittance Thermal Suite	GSFC/JHU-APL, Sensortex, Ashwin-Ushas
	Cold Gas Micro Thruster	Marotta Scientific Controls
	Ultra Low Power (1/4 V Logic)	GSFC/UNM-Microelectronics Research Center
	Miniature COMM Components	AeroAstro
Constellation Technologies	A Formation Flying and COMM Instrument with GPS (CCNT)	JPL
	Software Tools for Autonomous Ground Operations	Bester Tracking Systems

The ST-5 mission, as defined in Table 2, will involve a constellation of three microsatellites (Figure 1) that will be launched to a geosynchronous transfer orbit (GTO) with a final orbit of approximately 185 km by 35891 km. The spacecraft will fly as a secondary payload on an evolved expendable launch vehicle (EELV). As such, the launch date is not firm and could realistically occur anytime in late 2003 or early 2004.

SPACECRAFT DESIGN

Each spacecraft is spin stabilized such that the spin axis is perpendicular to the sun. Eight solar array panels are body mounted as shown in Figure 2. The panels, 28 cm

[1] Achieved using an Energetic Particle Detector, a Magnetometer, and a Sun Sensor.

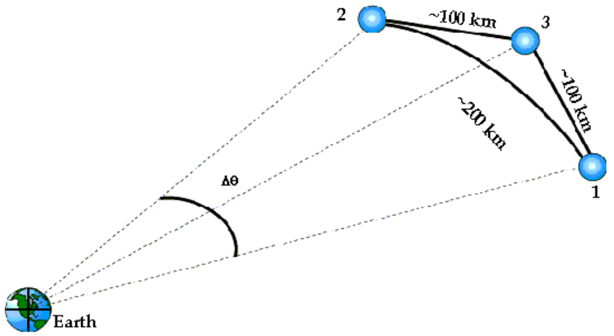

Figure 1 Baseline Constellation Configuration at Apogee

x 17cm x 0.5 cm, are aluminum honeycomb with graphite composite facesheets. The facesheets are 0.0254 cm thick. The structure is octagonal and is 28 cm in height by 46 cm in diameter. The spacecraft sides are 0.0813 cm thick aluminum sheets. The top and bottom decks are aluminum honeycomb (0.5 cm thick) with aluminum facesheets (0.038 cm thick). The card cage, a four-sided aluminum box with a wall thickness of 0.16 cm, is bolted near the middle of the spacecraft. It is used to house the power system and command and data handling electronics, which dissipate approximately 6 watts and 14 watts, respectively. 15.2 cm wedgeloks are used on each side of the cards to heat-sink them to the card-cage structure. The card-cage in-turn is bolted to the top and bottom decks using #6 and #10 screws.

Table 2 ST-5 Mission Characteristics

No. of Satellites	3
Mission Orbit	
Perigee	185 km (min)
Apogee	35891 km (max)
Inclination	0 to 28.5°
Period	10.5 hrs
Eclipse Duration	35 Minutes (perigee eclipse) 113 Minutes (apogee eclipse)
Mission Lifetime	Minimum 3 months, goal of 6 months
Spin Rate	20 rpm
Launch System	Atlas V or Delta IV
Orbit Adjust	Cold Gas System
Mass	< 25 kg
Size	46 cm Diameter x 28 cm High
Power (S/A BOL)	~23 Watts
Data	Max of 100 kbps
Attitude Control	Spin Stabilized, with spin axis perpendicular to sun
Control:	±5° (spin axis pointing control, 3σ)
Knowledge:	±1° (spin axis pointing knowledge, 3σ)
Communications	X-Band

With the exception of the passive nutation damper, which is mounted to the side of the spacecraft, all internal components are mounted to the top and bottom decks. All but four of the internal components are shown in Figure 2. Three of the four not shown are

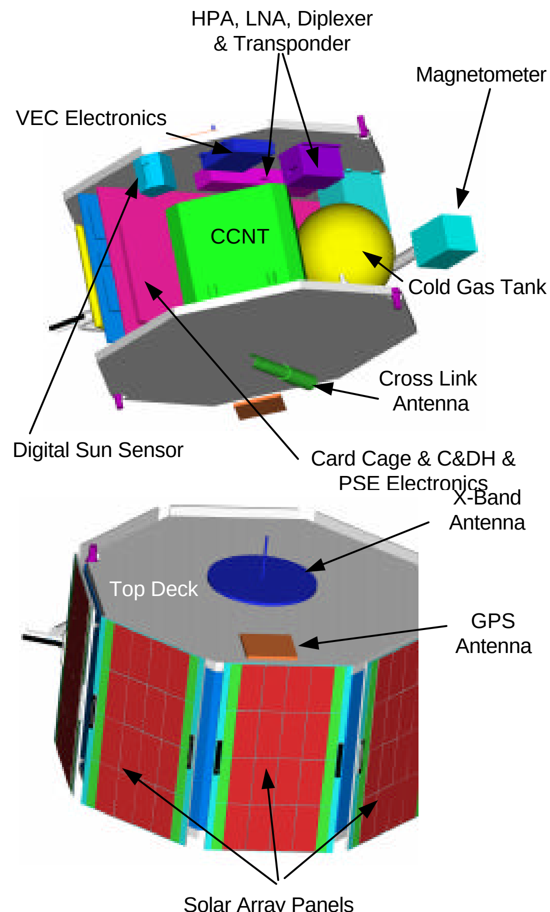

Figure 2 Internal and External Views of ST-5 Spacecraft

NMP technologies: the cold gas micro-thruster, the lithion-ion battery, and the flexible harnessing.

In addition to the balance masses (used to balance the spacecraft) and attachment points (for the deployer structure), components on the exterior of the spacecraft include:

1. The magnetometer mounted at the end of a 71.5 cm deployable boom, which is constructed of graphite composite
2. An X-Band antenna mounted on the top deck along the spin axis
3. One GPS antenna mounted to the top deck
4. A cross-link antenna mounted at the end of an 18 cm deployable boom constructed of graphite composite
5. Not shown in Figure 2 are two secondary radiators, each with a special coating that can vary its effective emissivity based on a voltage signal from an electronic controller (one on the top deck and one on the bottom deck)[2]. In addition, portions of

[2] Such variable emittance coatings can enhance the performance of a thermal design by saving heater power, increasing margins, and making the design somewhat more robust. While generically useful for all spacecraft, such coatings are especially suitable for very small spacecraft that are severely mass and power limited. The implementation and

the top and bottom decks are used as passive radiators for thermal control.

TCS TOP LEVEL REQUIREMENTS

The Thermal Control System (TCS) has to provide several functions. Firstly, the TCS is responsible for maintaining component and subsystem baseplate temperatures within survival and operational limits for all modes of operation. Secondly, the thermal design has to establish an acceptable temperature level for the interior of the spacecraft. Thirdly, it must control the heat flow for the integrated spacecraft and instrument interfaces. In addition to meeting the temperature and heat transfer requirements, the thermal control system must integrate the NMP new technology element, the variable emittance coatings (VEC), as functioning radiators while mitigating any risk impact due to their possible failure [Ref 1, 2, 3, 4, 5]. Finally, the TCS has to conform to mission-specific requirements and constraints. For example, it must comply with launch vehicle safety guidelines, surface charging, blanket grounding, heater power consumption, mass, etc. To ensure that the aforementioned requirements have been met, the TCS performance will be verified via analysis and spacecraft-level thermal vacuum and thermal balance testing. Prior to the manifestation of ST-5, Goddard created a study team whose charter was to design a nanosatellite for future missions where the apogee altitude could be as high as 40 Re. This type of orbit could result in harsh thermal environments. As a result, several thermal design concepts were investigated.

THERMAL CONFIGURATION TRADE STUDY

This section contains a synopsis of the three configurations considered. Because the study was conducted about a year prior to the inception of Space Technology 5, each concept was evaluated to assess general design robustness and effect of earth shadow on the design. The goal of the study was to develop a design that would keep the interior of the spacecraft warmer than -20°C at the end of the eclipse. The results presented indicate basic features of each design strategy and helped to guide a thermal design, but do not reflect individual component temperatures due to the simplified nature of the model used to generate the presented information. At the time of the study, the maximum earth shadow for ST-5 was not known, therefore a shadow time of 8 hours was considered.[3] However, the effect of the shorter eclipse is addressed at the end of this section. In addition, the study was completed for a 10-kg spacecraft rather than a 20-kg spacecraft, which would also affect the rate of temperature drop.

analysis of the variable emittance coatings are beyond the scope of this paper and will be addressed in a future publication.

[3] Future nanosat missions will likely have apogee eclipses lasting up to 8 hours as satellites are launched out to 40 Re and beyond.

The three thermal configurations considered were as follows:

1. Top and bottom of the spacecraft insulated with blanketing on the top and bottom decks. The inside of the cylindrical solar array not insulated allowing heat transfer between the internal equipment and the solar array.
2. The entire spacecraft insulated, top and bottom as well as inside the solar arrays, except for a radiator on top sized to radiate the internal electrical dissipation.
3. The entire spacecraft insulated with blanketing on the inside surface of the spacecraft bus, resulting in the internal equipment thermally isolated as well as possible from an "outside shell". A controllable two-phase heat transport device that can be "shut off" during earth shadows thermally couples the equipment to a radiator on the outside surface.

CONFIGURATION 1: UNINSULATED SOLAR ARRAYS - For configuration 1, where only the top and bottom are insulated with multilayer insulation (MLI), the basic in-sun temperatures are set by the thermal optical properties (solar absorptance and emittance) of the solar cells and can be adjusted, if necessary, to near room temperature with a relatively small radiator area on the top and bottom decks of the spacecraft. The key advantage of this configuration is its reliability, or robustness. Since the temperature of the spacecraft is set by a high energy balance (heat in = heat out) dominated by the absorbed solar energy, the operational temperature of the spacecraft is relatively insensitive to top and bottom MLI properties, or, largely, to internal heat dissipation.

However, the feature that yields the operational reliability, i.e., the high-energy balance, also results in a rapid drop in temperature when the solar load disappears during the earth shadow. During a 475-minute eclipse used for study purposes, internal temperatures dropped by about 60°C, which would result in internal temperatures in the range of -30 to -40°C. At the same time, the solar arrays dropped to a temperature of about -60°C.

Even with an 8 hours eclipse these end-of-eclipse temperatures are reasonable, at least as survival temperatures, for some spacecraft components. The solar array temperatures at the end of the eclipse are not a problem for standard cells. These temperatures, nonetheless, were too extreme for other components such as the battery.

CONFIGURATION 2: INSULATED WITH SIZED RADIATOR - Configuration 2 is fully insulated except for a passive radiator sized to yield operational (in-sun) internal temperatures near room temperature. The insulated nature of the design results in a much smaller overall energy balance than configuration 1. As a result, this configuration is much more sensitive to MLI properties and to internal power dissipation than configuration 1. However, as expected, eclipse performance improves.

During the ~8 hour eclipse, internal temperatures drop much less, i.e., by only about 20°C. This is a marked improvement with end-of-eclipse temperatures well within the range of most spacecraft components. It should be noted that the solar arrays, since they are now isolated from the body of the spacecraft, drop to temperatures of about -110°C. These temperatures are not necessarily problematic since solar arrays typically cycle over wide temperature ranges. For instance arrays in a LEO orbit swing between 90 and -80°C, while cells in a geosynchronous orbit typically swing between 90 to -160°C.

CONFIGURATION 3: MINI-CPL OPTION - The key feature of configuration 3 is that the internal equipment is completely isolated, both radiatively and conductively, from the outside "shell". The equipment is coupled to an external radiator with a two-phase heat transport device, such as a capillary pumped loop (CPL) or loop heat pipe (LHP) [6, 7, 8, 9, 10, 11]. Due to the temperature controllability inherent in two-phase systems and with properly sized radiators, the interior spacecraft temperatures are again maintained around 20°C. The CPL/LHP acts as a thermal diode to allow heat rejection when in the spacecraft is in normal operation and acts as a diode when it is in shadow. However, the temperature is also totally dependent on the proper operation of the two-phase loop. The two-phase heat transport device can be made redundant by the addition of a second loop if single fault tolerance is desired, but this adds mass. Note that redundancy is not a consideration for the other two configurations studied.

During the ~8 hour eclipse, a further improvement is realized, with internal temperatures dropping by as little as 6°C, if the internal payload is well insulated from the exterior of the spacecraft. As in configuration 2, the internal temperatures are sensitive to the MLI effectiveness and whether the internal equipment is conductively coupled or isolated from the bottom of the spacecraft. As in configuration 2, the solar array temperatures drop to about –110°C. For certain equipment or science instruments, the temperature control afforded by this type of "active" design may be necessary. It should be noted, however, that this type of design would impose an additional weight penalty.

Also note that it has been conservatively assumed in this study that no power is available during the eclipse, either for heaters to maintain temperature control, or to keep the loop "shut down" for configuration 3. Since the current state of the technology requires utilization of a small amount of power to "shut down" two-phase systems, this would constitute a technology development area. In addition, the small size and heat transport requirements of a nano-satellite application are also CPL technology development areas which are being pursued, leveraging ~15 years of CPL development at GSFC and recent successful tests of a small, cryogenic two-phase CPL.

DEPENDENCE ON SHADOW DURATION - The shorter (~2 hour) shadow expected for ST-5 would obviously result in smaller temperature drops during the eclipse periods, which made configuration 1 a viable solution. However, because the variable emittance coatings were sized such that the spacecraft was not dependent upon their performance, this particular configuration minimized the effectiveness of the variable emittance coatings. As a result, further studies were completed to evaluate a design that combined configurations 1 and 2. The baseline design is presented in the following sections.

ST-5 THERMAL SUBSYSTEM

OVERVIEW – As mentioned previously, the current design is a combination of configurations one and two. The preliminary thermal design of each small ST-5 spacecraft remains passive, consisting of electrically conductive thermal coatings, multilayer insulation (MLI) blankets, temperature sensors, conductive thermal isolators and thermal interface fillers. Except for a 200 cm^2 radiator, the top and bottom decks will be insulated with 18 layer multilayer blankets. In addition, to increase the energy balance of the system and keep the end-of-eclipse solar array temperatures warmer, blanketing was not used between the solar arrays and the spacecraft. The solar arrays are, however, conductively isolated from the structure

SIZING – The thermal design was established with the assumed condition that the spacecraft is spinning with the spin axis of the spacecraft normal to the ecliptic plane ± 5°. This includes the mission phase on the launch vehicle prior to release, although short duration excursions from this design attitude can be tolerated. The thermal design was also sized assuming an instantaneous spacecraft internal heat dissipation of 25 watts (solar array power plus battery) in full sun and 7 watts during eclipse. The passive radiators were sized to allow heat rejection from the components mounted to those surfaces, while keeping the end of eclipse component temperatures above –10°C. The variable emittance radiators were sized such that if they failed in the high emittance state, the internal spacecraft temperatures would at no time drop below –10°C. Conversely, if the variable emittance coatings failed at the low emittance state, the internal spacecraft temperatures would not increase above 40°C at anytime during the mission.

ENVIRONMENT - One of the major TCS design parameters is the spacecraft on-orbit thermal environment, which is defined by the spacecraft's orbit and attitude. The spacecraft will be placed in a highly elliptical orbit with a perigee altitude of approximately 185 km and an apogee altitude of approximately 35791 km. The spacecraft orbit inclination will be between 0 and 28.5°. Since the spacecraft travels very slowly near apogee, if the spacecraft is in the earth shadow during this time, it could be in an eclipse for up to 113 minutes. Depending on the launch date, during the 3 month mission the spacecraft could experience no eclipse or a perigee eclipse as short as 35 minutes. In any event, the Thermal Control Subsystem (TCS) has to be designed to meet temperature requirements for both extremes.

Table 3 Component Temperature Requirements

	Operating (°C)	Survival (°C)
Sun Sensor	-20 to 50	-40 to 60
Nutation Damper	-20 to 50	-40 to 60
C&DH Electronics	-20 to 50	-40 to 60
CCNT	-10 to 40	-30 to 60
Communication Units	-20 to 50	-40 to 60
Antennas	-60 to 40	-80 to 80
Battery	-10 to 40	-10 to 40
Power System Electronics	-10 to 40	-20 to 50
Solar Array Panels	-50 to 55	-75 to 70
Flexible Harness	-50 to 55	-75 to 70
Propulsion System	-20 to 40	-40 to 50
VEC Radiators	-40 to 40	-60 to 60
VEC Electronics	-20 to 50	-40 to 60
Actuators	-55 t0 50	-80 to 70
Spacecraft Bus	-50 to 40	-70 to 60
Instruments	-20 to 40	-30 to 50

TEMPERATURE LIMITS - It is the function of the ST-5 TCS to maintain all spacecraft components or interfaces within their applicable temperature limits once integrated into the spacecraft. Presented in Table 3 are the component operating and non-operating (survival) temperature limits. These limits are defined as follows:

- Operating: Project approved mission temperature limits within which the component operates satisfactorily.
- Survival (non-operating): Project approved mission temperature limits within which component will survive without damage and operate satisfactorily when returned to its operating temperature limits.

MASS - Minimizing thermal control hardware mass is extremely important given the small size of the ST-5 spacecraft. The mass allocation for the thermal control system, excluding the VEC radiators and control electronics, is 760 grams per spacecraft.

POWER - No heater power has been allocated to the thermal control system to maintain spacecraft temperatures during eclipse, launch or early orbit.

RELIABILITY - The reliability of the design of the ST5 TCS is consistent with a useful orbital design life of not less than 3 months and a goal of 6 months. The design items have been specified such that performance will be maintained considering all identifiable wearout factors and expendable depletions, and to minimize or eliminate potential sources of human-induced failures.

COATINGS - In addition to a traditional “yellow” paint for thermal control purposes, a portion of the total radiator area will include a new technology element, which is a specialized coating that can change its effective emissivity in response to variable sink and/or thermal load conditions. All interior components will have their heat transfer augmented by coating their outer surfaces with both a high emittance and high absorptance thermal coating (≥0.85).

HEAT TRANSFER - The primary thermal path for all components is by conduction through their baseplate and sidewalls to their mounting interface (spacecraft). Some components will utilize thermal fillers between their mounting plate and the spacecraft to minimize the resistance between the two. Those components, such as the Transponder, requiring high heat transfer with electrical isolation will utilize CHO-THERM 1671. Components that have no electrical resistance requirement will utilize Q-Pad 3.

THERMAL ANALYSIS

Table 4 Cold & Hot Case Assumptions

	Cold	Hot
Eclipse	1.88 hr	No
Earth Albedo Factor	0.25	0.35
Solar Constant	1285 W/m^2	1418 W/m^2
Planet Power	240.9 W/m^2	230.4 W/m^2
Power Profile	20 watts in sun, 7 watts in eclipse or EOEC	20 watts or DON
MLI Emittance	0.1	0.03
VEC Emittance	0.8	0.2

This analysis was prepared during the early to middle part of the spacecraft formulation phase. During this period, design trades and parametric studies were still underway. In addition, launch details were unknown. As a result, the purpose of the analysis conducted was to evaluate the various designs scenarios and bound the worst case conditions. In addition, it did not include launch and ascent.

Table 5 Thermal Dissipations (Watts) for Early Orbit Eclipse Case (EOEC)

Component	Geocentric Distance (Re)												
	1.03	**2**	**3**	**4**	**5**	**6**	**6.6**	**6**	**5**	**4**	**3**	**2**	**1.03**
Magnetometer	1	1										1	1
EPD													
GN&C (Sun Sensor)	0.1	0.1	0.1	0.1	0.1				0.1	0.1	0.1	0.1	0.1
S/C CPU	4	4	4	4	4	4	4	4	4	4	4	4	4
Power	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02
Thermal	0.5	0.5	0.5	0.5	0.5	0.65	0.65	0.65	0.5	0.5	0.5	0.5	0.5
Rcvr	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5
Xmtr	5.5	5.5	5.5	5.5	5.5				5.5	5.5	5.5	5.5	5.5
Propulsion	1.2				1.2		1.2		1.2				1.2
CCNT													
CCNT Sleep													
Total Power (Watts)	14.82	13.62	12.62	12.62	13.82	7.17	8.37	7.17	13.82	12.62	12.62	13.62	14.82

Table 6 Thermal Dissipations (Watts) for Day Ops Nominal (DON)

Component	Geocentric Distance (Re)												
	1.03	2	3	4	5	6	6.6	6	5	4	3	2	1.03
Magnetometer	1	1		1	1	1	1	1	1	1		1	1
EPD				1	1	1	1	1	1	1			
GN&C (Sun Sensor)	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1
S/C CPU	4	4	4	4	4	4	4	4	4	4	4	4	4
Power	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02	0.02
Thermal	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5
Rcvr	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5	2.5
Xmtr		5.5	5.5	5.5						5.5	5.5	5.5	
Propulsion	1.2												1.2
CCNT	9				9				9				9
CCNT Sleep		3	3	3		3	3	3		3	3	3	
Total Power (Watts)	18.32	16.62	15.62	17.62	18.12	12.12	12.12	12.12	18.12	17.62	15.62	16.62	18.32

Design margin was implemented by using conservative hot and cold case modeling assumptions to overestimate the predicted temperature extremes. Table 4 shows the assumptions that were used for both the cold and hot cases[4]. In general the worst case environmental conditions, power profiles, and optical properties were stacked. Initially, it was assumed that the components dissipated 25 watts (available array power and beginning of life plus 10%) while in the sun and 7 watts (battery output over 2 hours) during the eclipse. However, as the design evolved, distinct power profiles, corresponding to operational scenarios, were developed. The one that resulted in the least amount of power, Early Orbit Eclipse Case (EOEC) was used for the cold case. Conversely, the one that resulted in the maximum power, Day Ops Nominal (DON) was used for the hot case. The actual thermal dissipations for both of these cases are shown in Tables 5 and 6.

MODEL – A geometric math model (GMM) was built in Thermal Synthesizer System (TSS) to calculate the view factors, radiation couplings, and absorbed fluxes on the external spacecraft surfaces and the radiation couplings between the internal spacecraft surfaces (Figure 3). Inputs to the model include surface properties and the

[4] Normally, end of life optical properties are used, however,during this phase the optical properties were not varied. Once the design is "finalized", the optical properties that yeild the maximum heat loss will be used for the cold case and the optical properties that yeild the maximum heat gain will be used during the hot case.

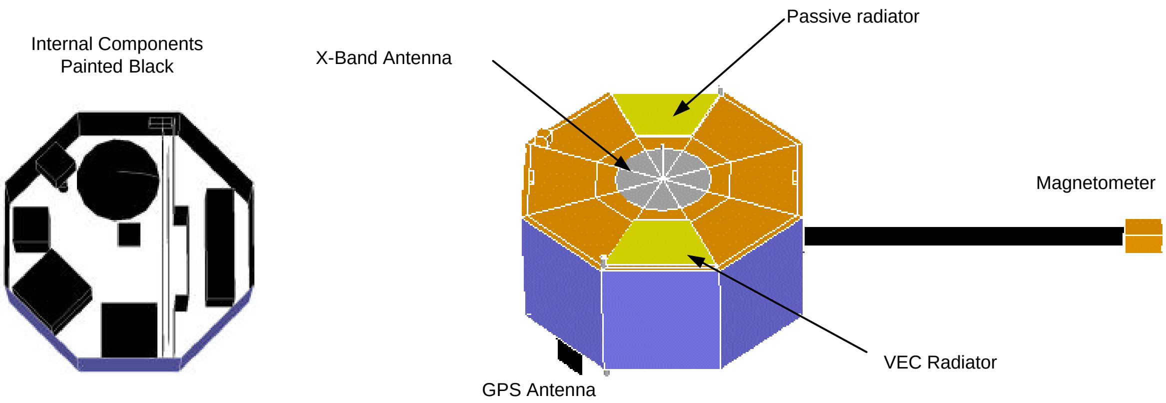

Figure 3 Geometric Math Model

orbit definitions for the hot and cold cases. TSS cannot import a mechanical drawing; as a result the geometry of the ST-5 was developed independently.

The spacecraft model had a total of 64 nodes. Although each component was modeled as a lump node, the physical representation of each was accurate to ensure correct view factors. Both decks were broken up into 16 nodes, while the solar arrays and sidewalls were modeled as eight identical panels. To cut down on the CPU time required to run the TSS model, the geometry was split up into external and internal models. The external model included the solar arrays, top and bottom decks, and the external components. The internal model included all the internal components as well as the spacecraft bus and solar arrays, but the surfaces facing space were made inactive.

Once the geometry was built in TSS, the Radk and Heatrate applications were run for the external models. Since radiation will not play a major factor in the energy exchange inside the spacecraft, radiation was assumed to be negligible. The Radk Application yielded radiation couplings and the Heatrate Application yielded environmental fluxes for the cold and hot cases. This data was then fed into the thermal math model (TMM) that was built in SINDA.

The TMM represents the thermal system as discrete lumped parameter nodes and was used to calculate transient and steady-state temperatures for various conditions. Inputs to the model included radiation couplings and absorbed fluxes from the GMM, conduction couplings, and time-varying internal power dissipations. To minimize the effect of the initial temperature assumptions, the SINDA model was run over 10 orbits. The temperature data was captured every 10 minutes so that the environmental effects were easily observed.

RESULTS

As mentioned previously, the initial configuration had the solar arrays isolated from the spacecraft structure using MLI and conductive isolators. This configuration resulted in solar array temperatures of -65°C at the end of the 1.88 hr eclipse and peaks of 50°C during perigee. Although absolute temperatures are acceptable for most solar arrays cell, the adhesive under consideration could not span that range. So, to achieve less of a temperature swing on the arrays and increase the robustness of the design, they were coupled to the spacecraft. To determine the limitation of that coupling, several parametric studies were completed.

Table 7 Effect of Conductive Coupling Between S/C & S/A for Hot Case

	0.07W/°C	**7W/°C**
Interior Max/Min	28.0/24.5	27.9/24.2
Sidewall Max/Min	27.7/24.3	27.6/23.9
Top Deck Max/Min	27.7/24.2	27.5/23.9
Bttm Deck Max/Min	27.7/24.3	27.6/24.0
S/A Max/Min	39.6/24.3	27.8/23.9

Table 8 Effect of Conductive Coupling Between S/C & S/A for Cold Case

	0.07W/°C	7W/°C
Interior Max/Min	7.1/-7.7	6.4/-20.4
Sidewall Max/Min	6.9/-8.3	6.1/-21.0
Top Deck Max/Min	6.8/-8.3	6.1/-21.0
Bttm Deck Max/Min	6.9/-8.0	6.1/-20.7
S/A Max/Min	21.0/-41.4	6.4/-22.0

Tables 7 and 8 show the effect the conductive coupling has on spacecraft temperatures for both the hot and

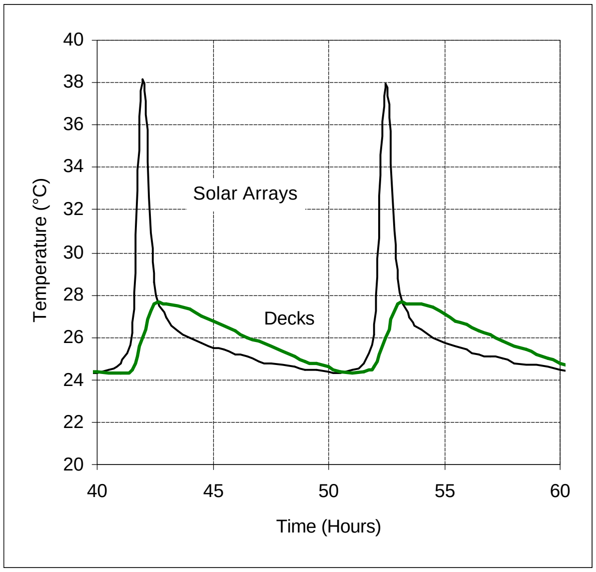

Figure 3 Temperatures for Hot Case

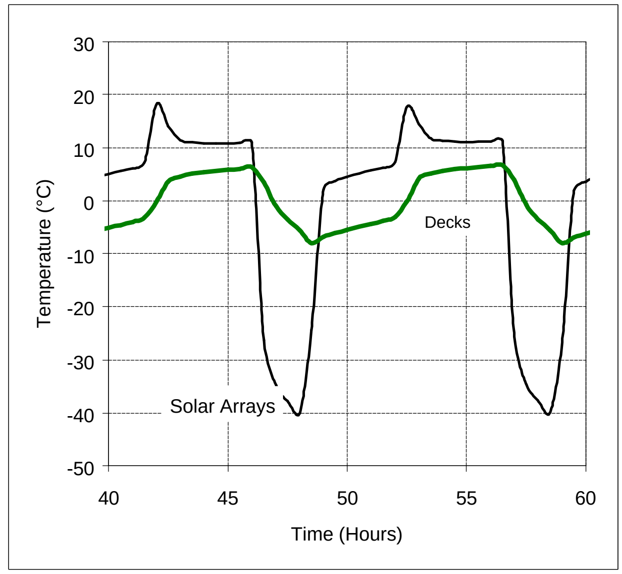

Figure 4 Temperatures for Cold Case

cold cases[5]. For these particular runs, the total radiator area (excluding the variable emittance coatings) was 400 cm^2 and a blanket effective emittance of 0.03 was used. The total power for the hot case (no eclipse) was between 12 and 18 watts, while the total power for the cold case was between 7 (during eclipse) and 15 watts (in sun). The variable emittance coatings were coupled to the spacecraft with a conductance value of 0.1W/°C, which allows them to function as spacecraft radiators without compromising spacecraft reliability.

As the conductive coupling between the solar array and the spacecraft was increased, both the spacecraft decks and the solar array reached a maximum temperature of approximately 28°C. At the end of the 1.88 eclipse, however, the interior temperature dropped to –20°C. These temperatures yield more than 10°C margin on the survival limits of most components except the lithium ion battery, which has a lower limit of –10°C. Additional analysis showed that by reducing the conductive coupling, the lower limit could be met. As expected, reducing this coupling resulted in warmer array temperatures for the hot case and colder array temperatures at the end of eclipse. Since the bulk of the mission will be spent in sunlight, it's improbable that the battery will see such cold temperature, however, other options for achieving warmer battery temperatures include isolating the battery or utilizing a small heater.

Figures 3 and 4 show temperature profiles for the hot and cold cases, 0.07 W/°C run, under the conditions defined in Table 4. Due to the small size of the spacecraft and low power dissipations, very little gradient is seen throughout the deck (less than 1°C). As a result, the two curves shown represent the average temperature for both the solar array panels and the spacecraft decks.

As illustrated by the graphs, the spacecraft temperatures spike at perigee when the spacecraft is closest to earth. Because of its relative small amount of mass, the solar arrays experience the largest temperature swing. In the hot case the solar array experiences a temperature swing from around 25°C to 40°C. Conversely, at the end of the eclipse, the solar array drops from approximately 20°C to –40°C. The spacecraft temperature swings are more benign and occur over a longer period.

CONCLUDING REMARKS

The initial top-level trade study considered 3 basic thermal architectures for the overall small spacecraft design. A relatively simple thermal model was built to examine the extreme hot and cold case for a nanosatellite spacecraft. Comparison of temperature requirements and results of a simple transient cool down analysis for the maximum 2 hr ST-5 eclipse indicate that coupling the solar arrays to the spacecraft and blanketing the top and bottom decks, leaving cutouts for radiators, results in a design that:

1. Is simple and therefore inexpensive, reliable, and easy to implement.

2. Is able to maintain identified temperature requirements with the spacecraft operating in sun or through a maximum 1.88 hour earth eclipse

3. Enables validation of the variable emittance coatings as a functioning radiator.

[5] Radiation coupling between the back of the solar array and the spacecraft sidewall was assumed to be constant. Each surface was painted with a high emittance, high absorptance coating.

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DEFINITIONS, ACRONYMS, ABBREVIATIONS

A: Amperes
ACS: Attitude Control System
AO: Atomic Oxygen
BOL: Beginning of Life
C: Celsius
CCNT: Constellation Communication and Navigation Transceiver
CGMT: Cold Gas Micro Thruster
cm: Centimeter
COMM: Communications
C&DH: Command and Data Handling
CPL: Capillary Pumped Loop
DON: Day Operations Nominal
DOSE: Day Orbit Science Event
EOEC: Early Orbit Eclipse Case
EOL: End of Life
EPD: Energetic Particle Detector
EPS: Electrical Power System
GMM: Geometric Math Model
GN2: Gaseous Nitrogen
GSFC: Goddard Space Flight Center
GTO: Geosynchronous Transfer Orbit
HPA: High Power Amplifier
kg: Kilogram
km: Kilometer
LEO: Low Earth Orbit
LHP: Loop Heat Pipe
MLI: Multilayer Insulation
NMP: New Millennium Program
PLF: Payload Fairing
PSE: Power System Electronics
Re: Earth Radius
S/A: Solar Array
S/C: Spacecraft
sq_cm: Square centimeters
ST-5: Space Technology 5
TCS: Thermal Control System
TMM: Thermal Math Model
TSS: Thermal Synthesizer System
VEC: Variable Emittance Coating
W: Watts
Xponder: Transponder